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PRELIMINARY PROCEDURES FOR CONTINGENCIES OCCURING DURING THE HIGH ELLIPSE PHASE OF MISSION E

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PROJECT APOLLO

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PRELIMINARY PROCEDURES FOR CONTINGENCIES OCCURRING DURING THE HIGH ELLIPSE PHASE OF MISSION E

by Bobbie D. Weber

SUMMARY

This paper presents the procedures to be followed if a contingency occurs during the high ellipse phase of Mission E (CSM-104/LM-4) which includes the translunar injection (TLI) maneuver through the simulation of the lunar orbit insertion (LOI) maneuver.

Sufficient trajectory data are provided to substantiate the procedures; however, the primary purpose of the paper is not to provide a trajectory analysis (which will be provided in the operational abort document) but to provide discussion of the abort philosophy for Mission E which has evolved from a series of meetings between FCD, FCSD, the flight crew, MPAD, and TRW.

INTRODUCTION

Approximately 3 hours 16.5 minutes after lift-off the S-IVB booster will be reignited to place the S-IVB/CSM/LM on an ellipse having a 3956-n.mi. apogee (h_a) and a 107-n.mi. perigee (h_p). Subsequently the CSM will perform transposition and docking (T&D) maneuvers and extract the LM just after passing the first apogee. At apogee of the second high ellipse, the SPS is ignited to raise perigee to 130 n.mi. The CSM/LM remains in the 3956- by 130-n.mi. orbit until approximately 14 hours 32.5 minutes g.e.t., at which time the SPS will be fired to simulate the lunar orbit insertion (LOI) burn and place the CSM/LM in a 400- by 130-n. mi. orbit.

Reference 1 gives a detailed description of the activities throughout the high ellipse portion and other phases of Mission E.

As indicated in reference 1 the sizing of the high ellipse was determined by navigation sighting requirements and SPS ΔV limitations.

The short SPS maneuver at second apogee, termed the midcourse correction (MCC), was sized primarily to meet tracking requirements for the subsequent LOI simulation burn. Figure 1, which has been extracted from reference 1, shows the ground tracks and indicates the sequence of events during the nominal high ellipse phase of Mission E. Note that perigee occurs at approximately 25° N latitude on the descending leg of each ground track.

In developing practical solutions to the abort problem for the high ellipse phase, it must be kept in mind that this portion of Mission E is unique to the Apollo Program; that is, no other nominal mission, as presently planned, will inject the spacecraft in a high-apogee, non-free-return ellipse. It is desirable to develop contingency procedures which are as consistent as possible from mission to mission and procedures which are similar to those for the nominal mission. Normally the course taken in developing contingency procedures is to develop procedures for the nominal mission trajectory, test the procedures on dispersed trajectories, and then compromise the procedures as necessary to gain the desired freedom in the degree of trajectory dispersions. Only during the TLI phase of Mission E do the prescribed contingency procedures bear any similarity to procedures being developed for subsequent Apollo missions. Thus, it was agreed that the primary method for returning the crew to earth in the event a contingency occurs during the high ellipse should bear some similarity to circularization and deorbit procedures developed for the nominal mission.

Two considerations have been presented which lead to the development of practical contingency procedures; contingency procedures should be developed to be consistent from mission to mission and if this is not possible, the procedures should bear some similarity to those for nominal events. Other considerations that should be noted are:

1. The procedure should be as simple as possible.

2. The procedure should allow for a high degree of error toleration.

3. Even though the contingency procedure can be considered a backup to the nominal courses of action, there should, if possible, also be a backup to the contingency procedure.

CONTINGENCY PROCEDURES

Procedures for the Translunar Injection Maneuver

As previously mentioned, the contingency procedures for the TLI phase of Mission E are very similar to the procedures being developed for subsequent Apollo missions. Reference 2 presents the preliminary contingency procedures developed for Mission F and subsequent missions. This section presents a cursory description of the procedures given in reference 2 and points out areas unique to Mission E.

The trajectory limits during the translunar injection are governed not by crew safety-related trajectory limits but rather by vehicle total attitude deviation and attitude rate limits which, if achieved, would require booster shutdown and a subsequent alternate mission to be performed. Reference 3 establishes the preliminary limits to be observed during the TLI and points out other crew monitoring considerations for TLI. Reference 4 shows the degree of trajectory deviations that can be expected for violating the total attitude deviation limits set forth in reference 3. Reference 4 also indicates that violations and booster shutdowns based on the total attitude deviation limits would never result in an earth intersecting ellipse. Therefore, trajectory limits such as excessive entry loads or insufficient time to entry will not be encountered prior to achieving the attitude deviation limits.

Should a contingency occur during TLI requiring the S-IVB to be shutdown and the immediate return of the crew to earth, the abort maneuvers for the last 140 seconds of TLI will be performed at a fixed time from S-IVB cutoff and at a fixed attitude with respect to the line of sight (LOS) to the far horizon (west of the subsatellite point). The abort maneuver will be targeted to the entry target line that is stored in the command module computer (CMC) return-to-earth abort program (RTEAP). This entry target line will be near the center of the operational entry corridor.

For aborts occurring during the first 10 seconds of S-IVB burn time (t_b), the procedure will differ. The procedures to be followed for the various burn times during TLI differ to allow landing to occur in water. Reference 5 investigated several different delay times from S-IVB cutoff and several different attitudes to determine a combination of delay times and attitudes that would provide the majority of the landing to occur in water. After evaluating the landing points, it was found that by providing one attitude and two delay times (applicable to different portions of TLI), land landings could be eliminated except for the first 10 seconds of S-IVB burn.

The following sections present summaries of the procedures for contingencies occurring at various times during TLI, requiring the immediate shutdown of the S-IVB and the immediate return of the crew to earth.

S-IVB burntime (t_B) < 10 seconds.- The CSM will separate from the S-IVB (procedures to be defined), the spacecraft will be tracked by the MSFN, a deorbit maneuver will be computed in the RTCC using the RTCC deorbit program, the deorbit maneuver will be performed in a heads-up attitude using a window marking located 31.7° from the X-body axis for backup attitude reference, and the subsequent entry will be performed using the EMS and the g meter in the constant g mode. To support this portion of the TLI burn preflight, the crew will be given one time to delay to SPS ignition and one abort ΔV for the abort maneuver computed at the normal heads-up deorbit attitude. This solution will be used in the event communications with the ground are lost and will result in landings near Hawaii (for the first TLI opportunity).

Table I shows the possible spread in landing points that could occur if the preflight data were used in lieu of RTCC deorbit computations.

10 seconds < S-IVB t_B < 60 seconds.- The procedure to follow during this period has been designated abort mode V. Following S-IVB cutoff, the CSM will separate from the S-IVB and then will orient to an attitude so that the thrust vector is aligned near the line of sight to the horizon (see ref. 6); the SPS will be ignited at 30 minutes after S-IVB cutoff; the abort maneuver will be performed in the "SCS auto" mode; and the subsequent entry will be flown in the backup mode (constant g entry).

Reference 6 describes in detail how the thrust vector is to be aligned for the modes V and VI abort maneuvers (to be discussed). Briefly, the crew optical alignment sight (COAS) reticle pattern will be used to align the thrust vector by aligning the -3° reticle marking on the apparent in-plane horizon. This procedure presupposes the COAS is mounted prior to TLI and that the reticle pattern is boresighted along the X-body axis. Aborts during this time period will be supported by providing the crew a chart of abort ΔV to apply at S-IVB cutoff plus 30 minutes as a function of inertial velocity at S-IVB cutoff. The same chart will also show the abort ΔV to apply for abort mode VI. The crew will determine the abort ΔV by selecting the V (inertial velocity), h (altitude), $\dot{h}$ (altitude rate) in the three display and keyboard (DSKY) registers prior to S-IVB cutoff. The TLI maneuver is a non-primary guidance navigation and control system (PGNCS) burn and is monitored by the CMC program P-47, designed for monitoring acceleration of a non-PGNCS burn.

P-47 allows the crew to call the V-h-$\dot{h}$ display at their discretion. The crew will record the V-h-$\dot{h}$ display corresponding to the time of S-IVB cutoff, and using this information in conjunction with the chart, they can determine the appropriate abort ΔV.

60 seconds < S-IVB $t_B \leq$ TLI ($\sim$ 150 seconds).- The procedure to follow during this time period has been designated abort mode VI, which is similar to mode V; the only difference is that SPS ignition will occur 10 minutes following S-IVB cutoff.

Figure 2 shows the expected magnitude of the modes V and VI abort maneuvers. Figures 3 and 4 indicate the landing sites associated with performing aborts in the modes V and VI regions, respectively.

Procedures for the High Ellipse Phase

Five possible methods for returning the crew to earth from the high ellipse were considered. The methods considered and the reasons for consideration are given below.

A. The first method involves circularizing the CSM/LM at perigee of the high ellipse using the service propulsion subsystem (SPS), CSM/LM separation, and deorbiting with the SPS from the near-earth ellipse. This method has been given prime consideration because the procedures involved parallel the procedures required in the nominal mission for the SPS to perform the LOI simulation and subsequently used to deorbit from a near-earth parking orbit. This procedure involves minimum training for both the crew and ground controllers as they would have received the necessary training while preparing for the nominal mission.

B. This method requires direct deorbit using the SPS. This method would apply to contingency situations requiring the return of the crew to earth as rapidly as possible while observing mission constraints, such as insuring that the CM enters within the entry corridor constraints, that sufficient time is available following the abort maneuver to prepare the CM for entry, and that landing occurs in water.

C. This method would require a SM reaction control subsystem (RCS) maneuver to be performed at apogee of the high ellipse to lower perigee to be within the entry corridor. This method would be required only in the event of an SPS failure.

D. This method would require a short duration LM descent propulsion subsystem (DPS) maneuver be performed to lower perigee altitude to not less than 75 n. mi. so that a subsequent RCS maneuver at apogee
456,000 FT.

could effect deorbit. This method was considered to serve in event the SPS failed and the perigee altitude had been perturbed so that the SM RCS alone could not effect deorbit.

E. The final method considered involves performing a SM RCS burn at apogee, separating from the SM, and subsequently performing CM RCS maneuvers to lower perigee to be within the entry corridor. This method was considered to serve in the event T&D was not performed or the LM had been jettisoned and the SPS failed. This method also assumes that perigee altitude had been perturbed so that SM RCS alone can not effect deorbit.

Both methods D and E would be used only in the event perigee altitude is perturbed so that the SM RCS alone can not effect deorbit. If it is assumed that the S-IVB achieves the nominal earth parking orbit, there are two possible instances when perigee altitude could be sufficiently perturbed to render the SM RCS incapable of effecting deorbit.

The first instance would be during the S-IVB TLI burn. If the TLI burn is performed within the limits prescribed in reference 3 or if the S-IVB is manually shut down following an overburn (5 to 10 seconds), the resultant perigee altitude will always be within the SM RCS deorbit capability. The next instance would be during the MCC at the second apogee of the high ellipse. To raise perigee to an altitude such that the SM RCS could not effect a successful deorbit (could not achieve the entry corridor), the SPS would need to overburn by approximately 1.5 seconds. The 1.5-second overburn required appears to be a very small error but the effect of the error (raises perigee approximately 7 n. mi.) is intolerable and should never occur.

Considering the negligible effect of TLI dispersions on perigee altitude and the fact that the MCC will be the first opportunity to determine the status of the SPS, it is unlikely that either method D or E will ever need to be employed to effect deorbit from the high ellipse. For these reasons a minimum of effort will be expended in developing the crew ground tasks involved in deorbiting using methods D or E. The development of procedures for method D is a strenuous flight planning task which has been performed at least preliminarily by FCSD. Also, it has been decided to use the procedures developed for Missions C and D for method E with only absolutely necessary changes. Preflight trajectory analysis and trajectory data prepared for the crew to carry onboard will not include data based on either method D or E.

For the sake of simplicity in recalling the appropriate procedures for the given methods of deorbit, to provide an abbreviated index to be included in the crew's flight plans, and to provide a curt description of the deorbit method, the methods have been designated type A through type E aborts.

It is presently planned to provide the crew preflight with abort solutions for types A, B, and C aborts for both the first and second TLI injection opportunities. One of the crew will catalogue the abort solutions in back of his flight plan with space provided for ground updates in real time. The solutions will be catalogued by placing all solutions for the first injection opportunity together and all solutions for the second injection opportunity together. All solutions for each type abort will be placed together and will then be arranged in alphabetical order. Each solution will be preceded by a code which will give abort type/injection opportunity/solution number. This code will then be entered in the flight plan containing the catalogued abort data by placing the code parallel to the mission time corresponding to the ignition time for the abort maneuver. The crew can then determine the appropriate abort solution by evaluating present time, time required to prepare for the abort maneuver, and the return time required.

Type A aborts.- Once the decision to abort from the high ellipse phase has been made, the type A abort procedures would require that preparations be made to perform the LOI maneuver early (during the next perigee pass) to place the spacecraft in a near-circular earth orbit. A coplanar horizon-reference deorbit would subsequently be performed with the SPS to achieve a water landing near a prime recovery area.

The nominal Lambert target vector selected for the nominal LOI maneuver and the associated time from ignition to target do not require updating for the early LOI maneuver.

The primary purpose in achieving an intermediate near-earth parking orbit before deorbiting during the high ellipse phase is to provide more landing site control or for the case of a coplanar deorbit, to provide freedom in selecting the landing longitude.

When single-impulse deorbits are initiated from the high ellipse, the aborts are targeted to an entry target line which is a function of the inertial velocity (V_i) and inertial flight-path angle (γ_i) at 400 000 ft. When a minimum entry time constraint (providing sufficient time to prepare CM for entry) of approximately 20 minutes from abort to drogue chute deployment is placed on the postabort trajectory, the majority of acceptable abort solutions (ref. 7) result in landings which cluster about a geographical midpoint (near the preabort perigee location) by about $\pm 3^\circ$ in both longitude and latitude. Therefore, for a given revolution during the high ellipse phase, although the SPS is capable of deorbiting from any point in the high ellipse, the resulting landing positions will be nearly the same.

Preflight solutions for the type A aborts will be provided the crew and flight controllers based on preabort and postabort tracking and an optimization of lighting conditions at landing. In real time the ground will be required to update the LOI ignition time and uplink this time and the Lambert targeting parameters to the CMC through program 27. The crew will then select the Lambert targeting program (P31) to verify the targeting parameters.

Subsequent to the LOI maneuver the ground and crew will determine (relative to the contingency which caused the abort to be initiated) if it is permissable to update the onboard deorbit maneuver solution. If the situation permits updating the onboard solution, the ground will compute the deorbit solution and uplink the external ΔV targeting parameters. The ignition attitude for the deorbit maneuver will be such that the crew will be in a heads-up attitude and the earth's apparent horizon will be aligned with a window marking 31.7° above the X-body axis to provide a back-up attitude reference. The deorbit maneuver will be performed using the PGNCS SPS thrusting program (P-40), and the subsequent entry will be flown in the back-up mode (constant g entry). If the deorbit maneuver cannot be updated, the crew will perform an SCS automaneuver using the onboard deorbit solution. The ignition attitude is achieved by aligning the earth's horizon on the window marking, and at ignition the thrust vector is fixed inertially for the subsequent deorbit maneuver. The entry following deorbit will be flown in the back-up mode (constant g entry).

Figures 5(a) through 5(c) (taken from ref. 8) show the landing loci following the deorbit maneuvers subsequent to LOI maneuvers performed at the second, third, and fourth perigee of the high ellipse (where first perigee is at TLI). The landing loci are shown as a function of the delay time from LOI shutdown to deorbit maneuver ignition. Table II shows the optimized solutions selected based on tracking, lighting conditions at landing, and landing location.

The data presented are for the high ellipses following the first TLI opportunity and were based on a heads-down attitude for the deorbit maneuver; the data should therefore be considered only representative of the procedures as discussed.

Type B aborts.- If a contingency arises during the high ellipse phase requiring the immediate return of the crew to earth and if the return time required does not permit a type A abort to be performed, a single SPS burn will be performed for direct deorbit from the high ellipse.

As a result of the cluster effect described in the previous section, two discrete points (way stations) on each orbit of the high ellipse have been selected at which the SPS deorbit will be performed. One way station has been selected approximately 12 minutes prior to apogee. The reason for selecting this station at 12 minutes prior to apogee rather that at apogee was to provide sufficient time to prepare for a split burn about apogee with the SM RCS in the event the SPS failed to ignite. The other way station was selected to be at approximately 285° true anomaly. The reason for selecting this position was that if the deorbit is performed closer to perigee of the preabort orbit, the resultant entry time would not allow the crew time to orient the CM to entry attitude prior to atmospheric capture. The primary reason for selecting the second way station was to provide a second deorbit opportunity located about halfway around the preabort ellipse from the first opportunity. The way stations are about 1.4 hours apart, or about half the orbital period.

The way station to be used for the abort will be dependent on when the abort decision is made and the crew preparation time required prior to performing the abort. If the abort decision is made prior to the first way station (12 minutes prior to apogee), and if the crew has sufficient time to prepare for the abort the first way station will be used; if not, the crew will not abort until about 1.4 hours later at the second way station. For this case regardless of whether the first or second way station is used, the CM would land at approximately the same geographic position at about the same local time. However, if the abort decision were made prior to the second way station but the crew did not have sufficient time to prepare for the abort, the abort maneuver would have to be delayed until near the next apogee. In this case, the CM would land approximately 45° west of where it would have landed if sufficient time had been available to perform the abort at the 285° true anomaly way station.

The abort maneuver for the type B aborts will be performed in the SCS auto mode. At the ignition attitude, the spacecraft will be oriented with the crew heads down and the earth's apparent horizon will be aligned with a window marking 31.7° above the X-body axis to provide a back-up attitude reference. One of the reasons for selecting the heads-down attitude is that the aborts near apogee do not provide a lighted horizon. If it is determined by the crew that the back-up attitude reference is needed and that the earth's horizon is not distinguishable, the ground can provide window-star patterns for the crew to orient to for the ignition attitude. The primary reasons for selecting the heads-down, rather than heads-up, attitude is that the heads-up attitude requires very large abort ΔV's, the return times are too fast, and the maneuver is very critical with respect to attitude errors, particularly the abort at the 285° true anomaly way station. Very small attitude errors for aborts at the 285° true anomaly way station in the heads-up attitude would result in entry points outside the entry corridor and insufficient time to make corrections to the resulting trajectory. The way station prior to apogee would be subject to the same errors but there might be time for a correction. The attitude is heads-down for that way station to eliminate the possibility of encountering a situation which might require a midcourse and to keep the procedures consistent throughout the high ellipse phase.

The obvious advantages in selecting discrete points in a given ellipse for the type B aborts are that the abort ΔV and the abort attitude (whether local horizontal, earth horizon reference, or with respect to an inertial platform) remain constant. The abort ΔV's for the high ellipse prior to the MCC will be about 145 fps for the way station prior to apogee and about 500 fps for the 285° true anomaly way station. After the MCC, the ΔV for the pre-apogee way station will be about 185 fps but the ΔV for the other way station will remain at 500 fps. Entry following the abort maneuver will be flown in the back-up mode (constant g entry).

Tables III and IV, which have been extracted from reference 9, present trajectory data for the type B aborts. Note that for each solution number, two time of ignition and associated trajectory data are given. The two times represent the interval over which the given abort ΔV could be applied with as much as $\pm 5°$ attitude error or ± 10 percent ΔV error and still effect a successful deorbit. The $V_i - \gamma_i$ at 400 000 ft will be within the entry corridor). The even-numbered solutions represent aborts simulated at the 285° True anomaly way stations, and the odd-numbered solutions represent aborts simulated at the pre-apogee way station.

<u>Type C aborts</u>.- The type C aborts provide a back-up means of returning from the high ellipse in the event the SPS fails during one of the planned SPS burns or if the SPS fails to ignite for either the type A or B aborts.

The type C aborts would be performed in the SCS auto mode with ignition occurring about 6 minutes prior to apogee of the high ellipse resulting in a split burn about apogee.

Because of the low ΔV capability of the SM RCS, the type C aborts will not use the earth's horizon as a back-up attitude reference. The backup reference for planned type C aborts will require that the ground provide star-window patterns to the crew for back-up attitude reference.

If the type C abort is employed to back up an SPS failure where the SPS failed to ignite for the type B pre-apogee abort way station, the crew can establish the correct attitude accurately by uncaging the body mounted attitude gyros (BMAG) at the horizon reference attitude and pitching down (thrust vector and X-body axis pitched up) about 30°. The type C aborts can tolerate about a 10° pitch error and still provide deorbit capability. (The V_i - γ_i at 400 000 ft will lie within the entry corridor). Following the type C abort maneuver, the entry will be flown in the backup mode.

Procedures for the Simulated Lunar Orbit Insertion (LOI) Maneuver

The prime procedures to follow during this mission phase should be fairly obvious. If an abort is required during the last orbit of the high ellipse and the crew does not have time to prepare for the type B abort at the 285° true anomaly way stion, the nominal LOI maneuver will be performed and an SPS deorbit will be performed at the first opportunity.

If the SPS should fail during LOI, the crew would have to wait in the resulting parking orbit until the parking orbit perigee position is located in the Atlantic Ocean. This wait period would be about 11.2 hours, regardless of when during LOI the SPS failed. For the second injection opportunity, this wait period can be reduced to about half by performing an SM RCS deorbit to the Indian Ocean.

CONCLUDING REMARKS

Preliminary procedures for contingencies occurring during the high ellipse phase of the E Mission have been presented. Changes to the procedures will be noted in the minutes of the Apollo Abort Working Group meetings and will be accounted for in the operational abort document.

Primary consideration has been given to aborts from the high ellipse resulting from the first TLI opportunity. Studies are in process to verify the applicability of the procedures to the ellipse resulting from the second TLI opportunity.

TABLE I.- FIXED ATTITUDE DEORBITS DURING THE FIRST 10 SECONDS OF THE FIRST OPPORTUNITY TLI BURN

Horizon reference attitude,[a] Ψ, deg	+31.7
Deorbit delta velocity, ΔV, fps	350
Delay time from S-IVB shutdown, T_D, min . . .	48.5

S-IVB burn time, T_B, sec	Landing longitude, λ, deg W	Landing latitude, φ, deg N	Daylight remaining at landing, hr	Ground elapsed time of landing, hr
0	185.9	18.4	5.45926	4.44494
5	205.1	26.8	3.81155	4.54765
10	218.4	30.4	2.27441	4.62122

[a]Referenced to far horizon.

TABLE II.- OPTIMUM DEORBIT MANEUVER SOULTION SUMMARY

Perigee pass at which the LOI maneuver is performed	Delay time from LOI cutoff to deorbit maneuver initiation, T_D, min	Deorbit ΔV, fps	Time from deorbit to entry, T_{AR}, hr	Landing area	Daylight (darkness) remaining at landing, hr
2	32	190	0.734	Mid Pack	0.529
2	107	375	0.867	West Pack	3.578
3	24	278	0.778	West Pack	2.026
4	48	218	0.588	West Pack	(12.926)

TABLE III.- TYPE B ABORT DATA FOR FIRST INJECTION OPPORTUNITY OF MISSION E HIGH APOGEE ELLIPSE PHASE

High ellipse	Abort type	Solution no. (a)	ΔV, fps	Ignition		G.e.t. of landing, hr	Landing[b]		Darkness remaining, hr	Postabort tracking				Time from ignition to landing hr (c)
				G.e.t., hr	True anomaly, deg		Long., deg	Lat., deg		No. of stations	Contact time, hr	No. of stations	Contact time, hr	
1	B	1	145	4.27	150	6.14	98.W	25.1N	(11.9)	6	1.66	2	.05	1.87
				4.70	180	6.15	99.4W	25.2N	(12.0)	5	1.24	2	.04	1.45
1	B	2	500	5.77	275	6.17	99.1W	25.3N	(11.96)	2	.20	2	.05	.40
				5.87	290	6.20	90.6W	21.8N	(11.25)	1	.10	2	.10	.33
2	B	3	145	7.09	150	8.96	140.3W	25.0N	(11.90)	5	1.61	--	--	1.87
				7.52	180	8.97	141.7W	25.4N	(12.0)	4	1.20	--	--	1.45
2	B	4	500	3.55	271	3.99	141.0W	25.0N	(11.9)	2	.19	--	--	.44
				3.64	288	9.03	133.6W	21.9N	(11.3)	2	.1	1	.06	.39
3	B	5	122 185 127	9.95	152	11.79	177.E	24.9N	(11.9)	2	1.23	1	.22	1.84
				10.35	180	11.80	175.6E	25.3N	(12.02)	2	.84	1	.23	1.45
3	B	6	500	11.38	270	11.83	176.5E	24.9N	(11.92)	1	.27	--	--	.45
				11.48	288	11.86	175.9W	21.6N	(11.27)	1	.18	--	--	.38
4	B	7	135	12.78	152	14.62	134.6E	24.8N	(11.91)	6	1.34	--	--	1.84
				13.18	180	14.63	133.1E	25.2N	(12.02)	6	.96	--	--	1.45
4	B	8	500	14.22	272	14.66	134.4E	24.6N	(11.87)	--	--	--	--	.44
				14.31	288	14.69	141.6E	21.5N	(11.27)	1	.02	--	--	.38

[a]Even attempts have daylight horizons; odd attempts have night horizons.

[b]Entry corridor is not exceeded for ±5° attitude error or ±10% ΔV error. The impact point is based on RTCC entry ranging polynominal which approximates half-lift range.

[c]Time of landing does not include drogue or main chute time and is based on RTCC polynominal calculations.

TABLE IV.- TYPE B ABORT DATA FOR SECOND INJECTION OPPORTUNITY OF MISSION E HIGH APOGEE ELLIPSE PHASE

High ellipse	Abort type	Solution no. (a)	ΔV, fps	Ignition		G.e.t. of landing, hr	Landing[b]		Darkness remaining, hr	Postabort tracking				Time from ignition to landing hr (c)
				G.e.t., hr	True anomaly, deg		Long., deg	Lat., deg		No. of stations	Contact time, hr	No. of stations	Contact time, hr	
1	B	1	145	6.17	172	7.75	96.4W	12.8N	(9.82)	5	1.35	2	.09	1.58
				6.32	182	7.75	96.5W	12.8N	(9.82)	5	1.20	2	.09	1.43
1	B	2	500	7.35	270	7.77	97.7W	13.3N	(9.91)	5	.30	--	--	.42
				7.43	291	7.79	92.2W	9.9N	(9.42)	4	.23	--	--	.36
2	B	3	145	9.01	173	10.58	138.9W	12.6N	(9.82)	4	1.29	1	.08	1.57
				9.17	183	10.59	139.0W	12.6N	(9.82)	4	1.39	1	.08	1.42
2	B	4	500	10.15	270	10.60	140.3W	13.2N	(9.91)	2	.18	1	.08	.45
				10.24	285	10.63	134.6W	9.6N	(9.40)	2	.10	1	.11	.39
3	B	5	185	11.67	160	13.41	177.5E	13.0N	(9.91)	4	1.05	1	.26	1.74
				11.97	180	13.42	176.9E	13.3N	(9.95)	4	.76	1	.27	1.45
3	B	6	500	13.02	275	13.45	178.8E	11.9N	(9.75)	2	.26	--	--	.43
				13.08	285	13.47	177.2W	9.3N	(9.39)	2	.21	--	--	.39
4	B	7	185	14.51	160	16.25	134.9E	12.8N	(9.91)	6	1.23	--	--	1.74
				14.81	180	16.26	134.2E	13.1N	(9.95)	6	1.00	--	--	1.45
4	B	8	500	15.33	270	16.29	134.4E	12.7N	(9.90)	--	--	--	--	.46
				15.9[illegible]	285	16.31	140.2E	9.1N	(9.39)	1	.04	--	--	.39

[a]Even attempts have daylight horizon; odd attempts have night horizon.

[b]Entry corridor is not exceeded for ±5° attitude error or [illegible] ΔV error. The impact point is based on RTCC entry ranging polynominal which approximates half-lift range.

[c]Time of landing does not include drogue or main chute time and is based on RTCC polynominal calculations.

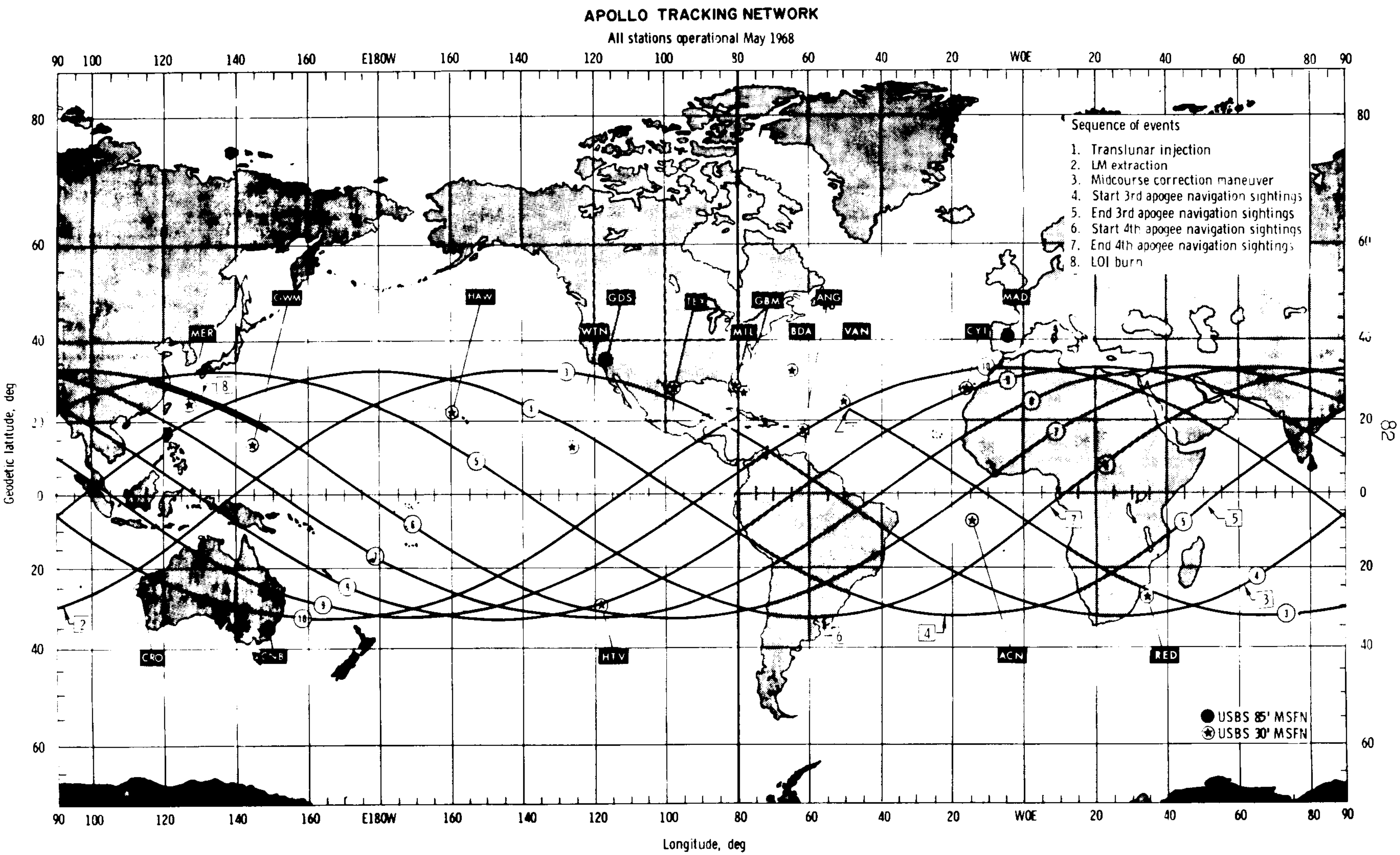


Figure 1.- Mission E ground tracks and major mission events (revolutions 3 through 10).

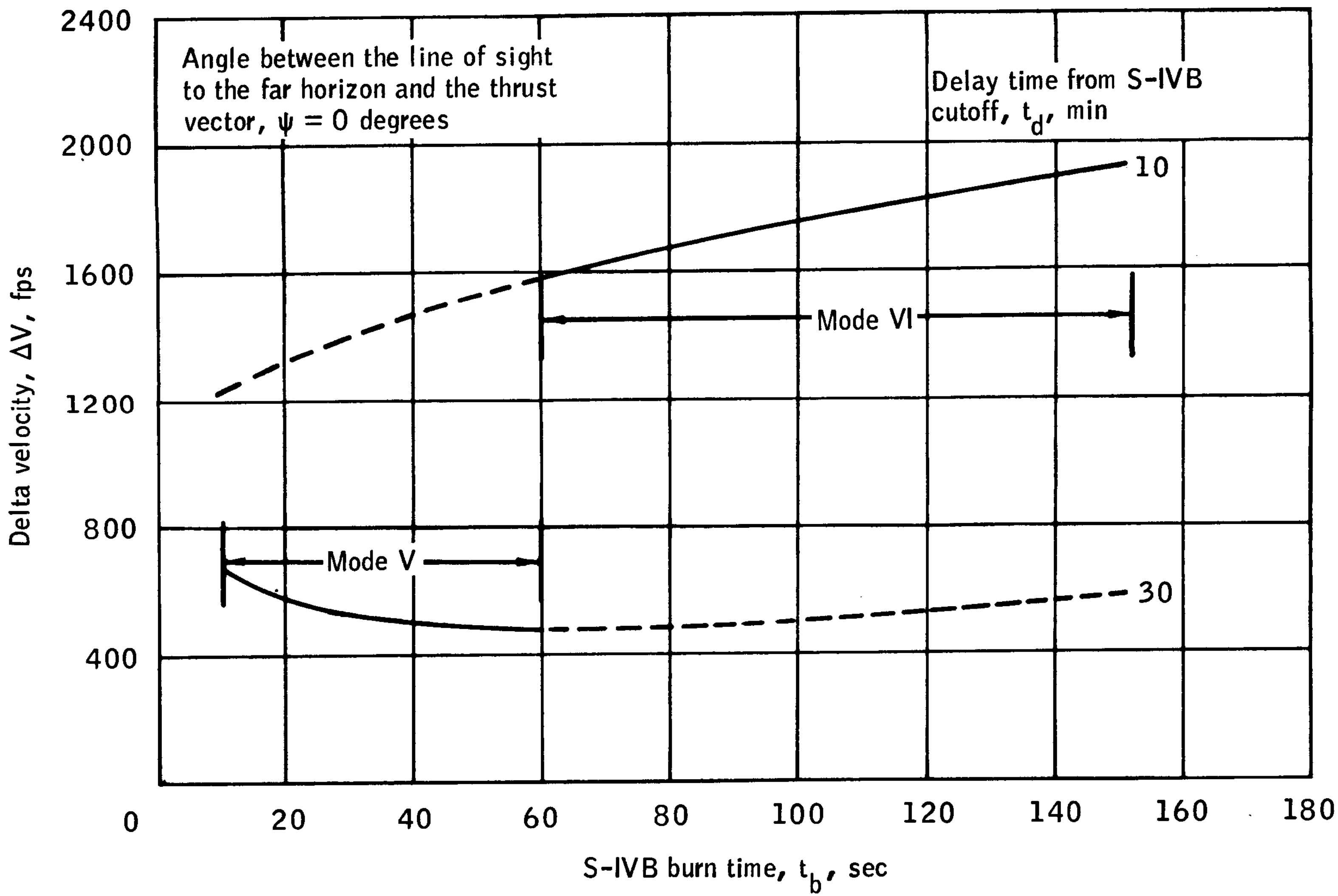


Figure 2.- Abort delta velocity as a function of S-IVB burn time and delay time from S-IVB cutoff.

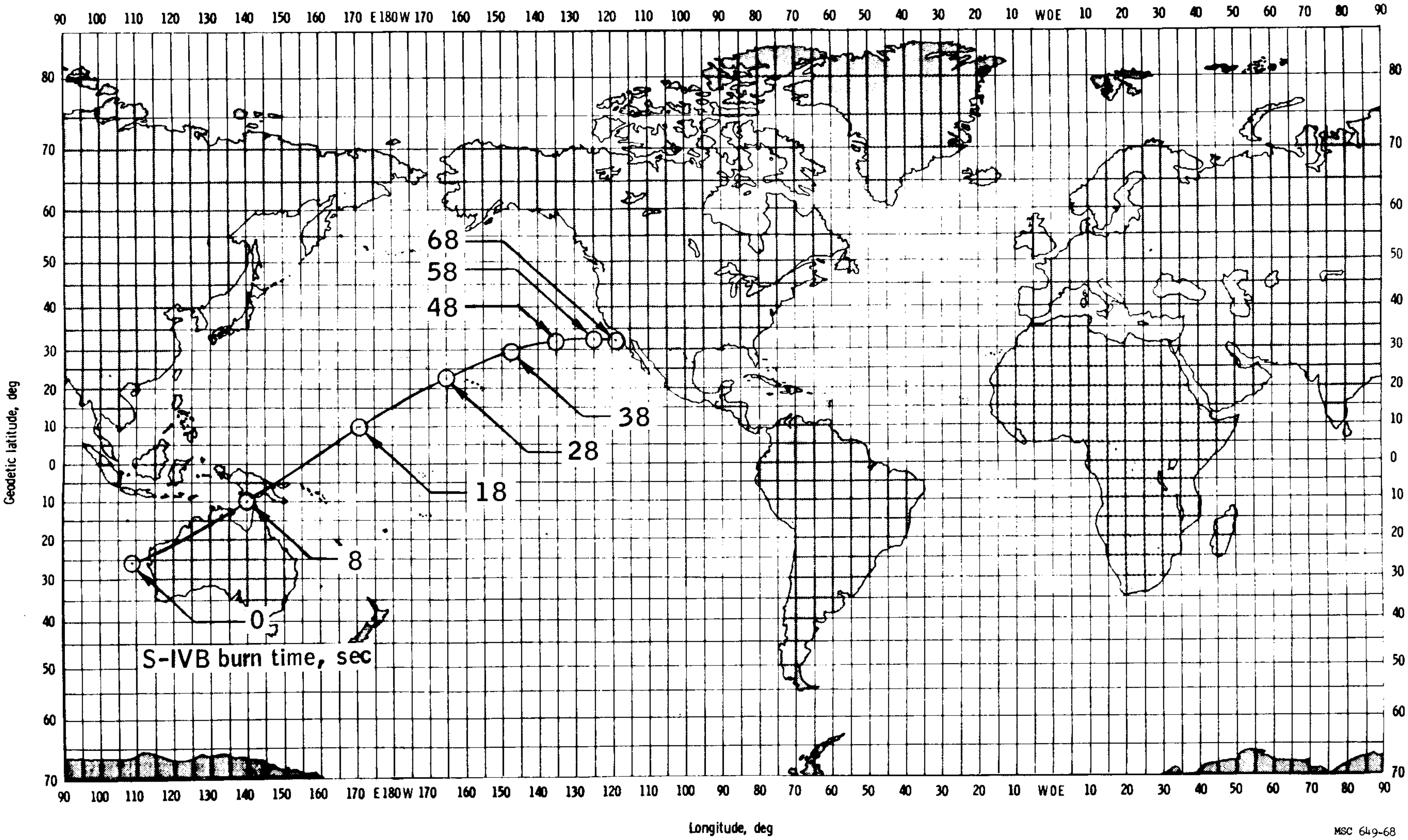


Figure 3.- Landing site loci as a function of S-IVB burn time for $\psi = 0°$ and $t_d = 30$ minutes.

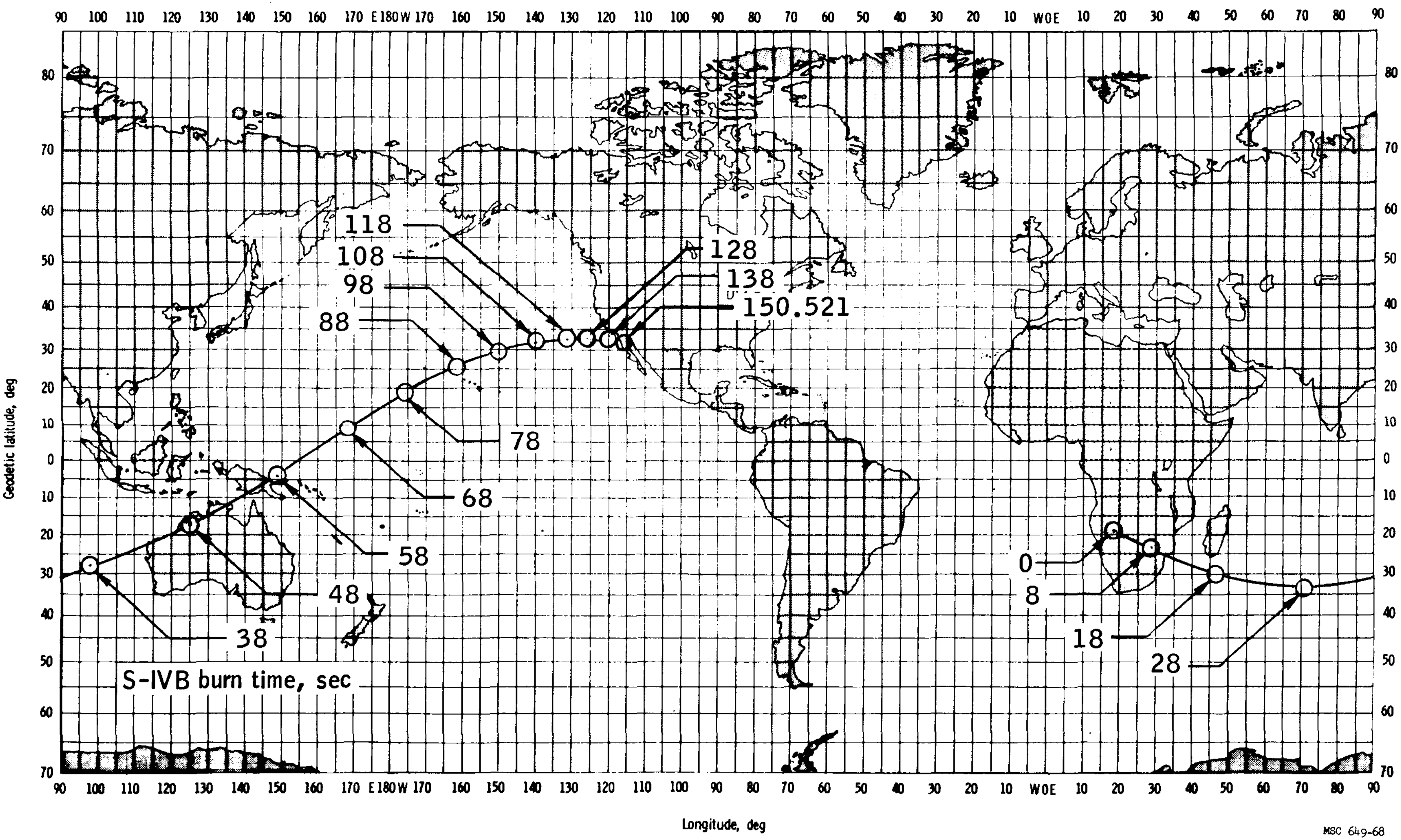
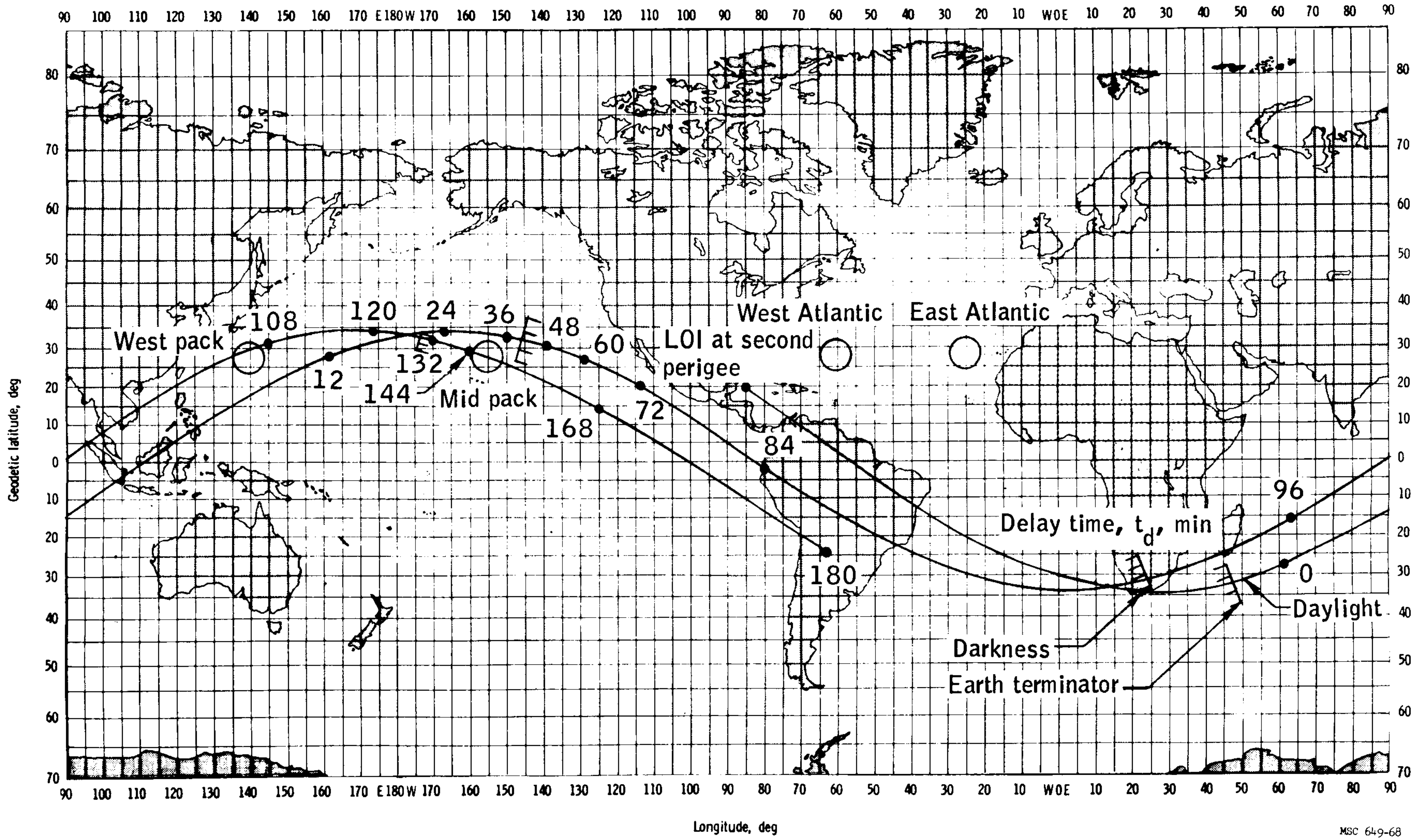
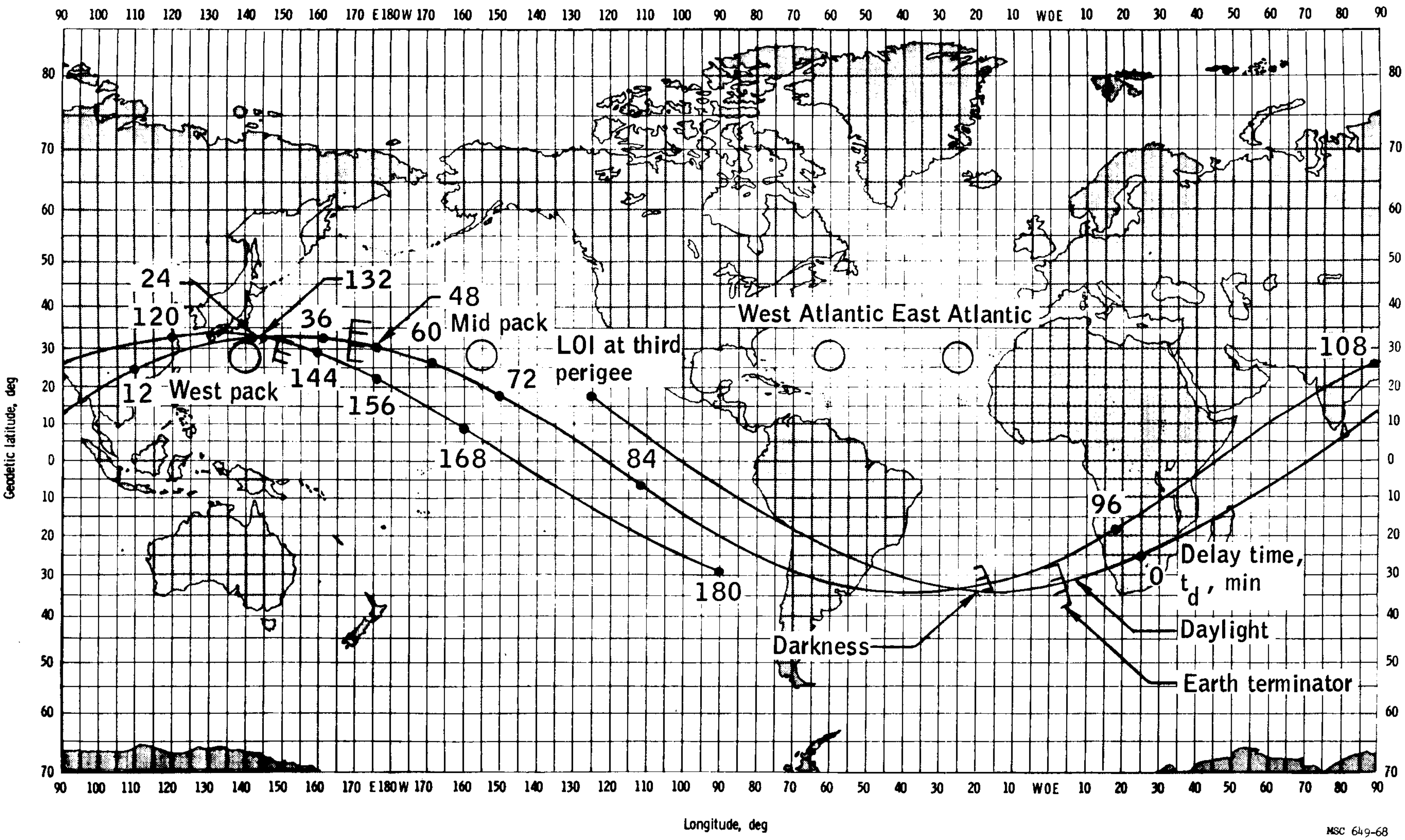


Figure 4.- Landing site loci as a function of S-IVB burn time for $\psi = 0°$ and $t_d = 10$ minutes.



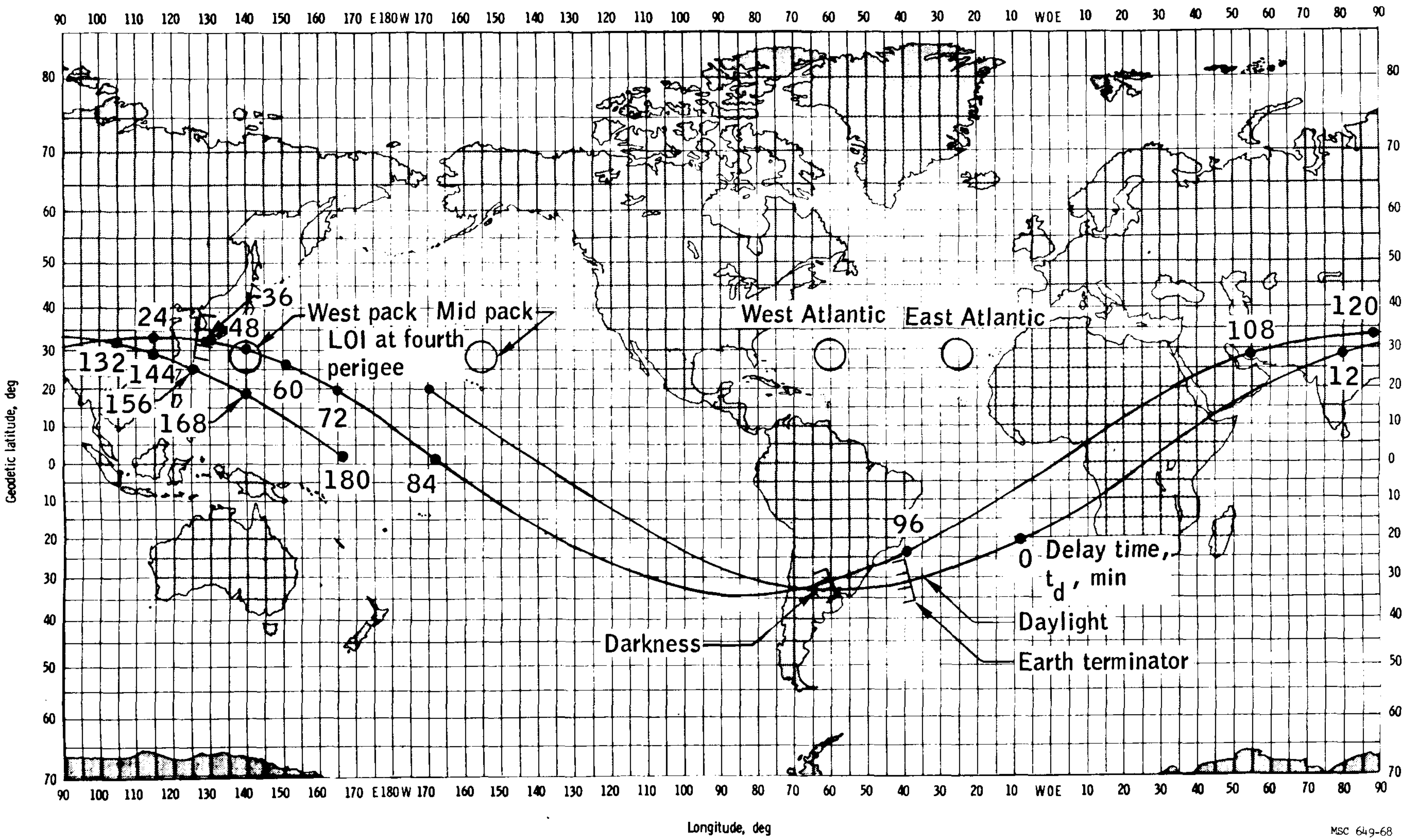
(a) Ground elapsed time of LOI = 06:02:59 hr:min:sec .

Figure 5.- Deorbit maneuver landing site loci as a function of delay time from LOI.



(b) Ground elapsed time of LOI = 08:52:40 hr:min:sec .

Figure 5.- Continued.



(c) Ground elapsed time of LOI = 11:42:35 hr:min:sec .

Figure 5.- Concluded.

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